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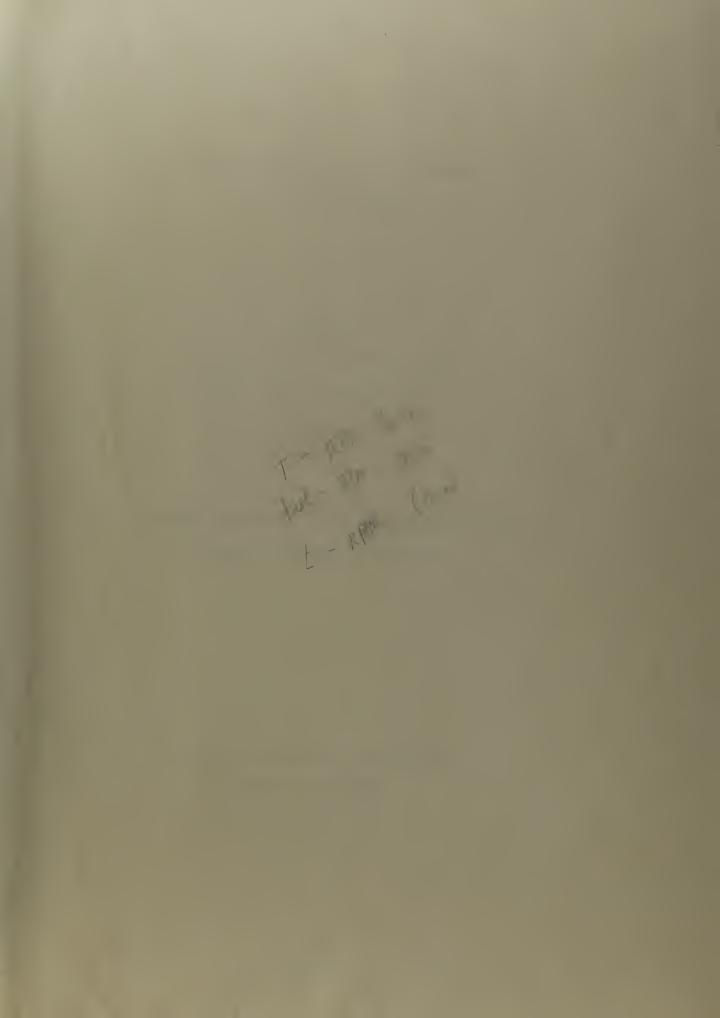
CALIFORNIA INSTITUTE OF TECHNOLOGY

THEORETICAL INVESTIGATION OF ACCELERATION
OF A TURBOJET ENGINE

Thesis by

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Lt. Cdr. John P. Wheatley, USN

In Partial Fulfillment of the Requirements for the Degree of Aeronautical Engineer

Pasadena, California

1947

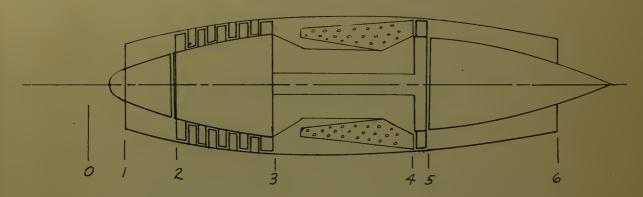
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AUKINO LEDGEMENT

The writers are sincerely grateful to Mr. W. D. Rannie for his assistance and consideration during the preparation of this report.



NO.JETULATURE



A ₁ , A ₂ , A ₃ , etc	Annular areas	ft ²
a ₁ , a ₂ , a ₃ , etc	Local sonic velocities	ft/sec
op, cv	Specific heats	ft - 1b slug - 0R
D _c	Compressor inlet Mean diameter	ft
Dt	Turbine mean diameter	ft
	Gross thrust	1b
g	Acceleration of gravity	ft/sec ²
I _D	Polar moment of inertia of rotor = 0.7675	1b - ft - sec ²
М	Mach number	
å	Mass rate of fluid flow	slu s/sec
n	Rotational speed	RPL
ñ	Rotor acceleration	Rev/sec ²



$P_{\mathbf{x}}$	Free stream pressure at station "x"	lb/ft ²
P _S	Stagnation pressure at station "x"	lb _{ft} 2
R	Gas constant	ft - 1b slug - R
T _X	Free stream temperature at station "x"	° _R
T _{SX}	Stagnation temperature at station "x"	° _R
Vx	Fluid velocity at station "x"	ft/sec
we	Jompressor power	ft - 1b sec
₩t	Turbine power	ft - 1b
Γο	oppressor flow coefficient	$\Gamma_c = \frac{\dot{m}}{P_2 A_c} \sqrt{\frac{R T_2}{2}}$
A.	Compressor power coefficient	$\Omega_c = \frac{W_c}{P_2 A_c a_z}$
6c	Jompressor speed coefficient	$O_c = \frac{O_c n}{a_2}$
Γ_{t}	Turbine flow coefficient	$ \Gamma_{t} = \frac{\dot{m}}{P_{4}A_{t}} \sqrt{\frac{RT_{4}}{2}} $
Ω_{t}	Turbine power coefficient	$\Omega_t = \frac{W_t}{P_4 A_t a_4}$
Ot.	Turbine speed coefficient	$\sigma_t = \frac{D_t n}{\alpha_4}$
8	Ratio of specific heats	C _P /C _v
ال د عال ، لا	Diffusor, burner, and nozzle efficiencies	
A6/A5	Tail area ratio	

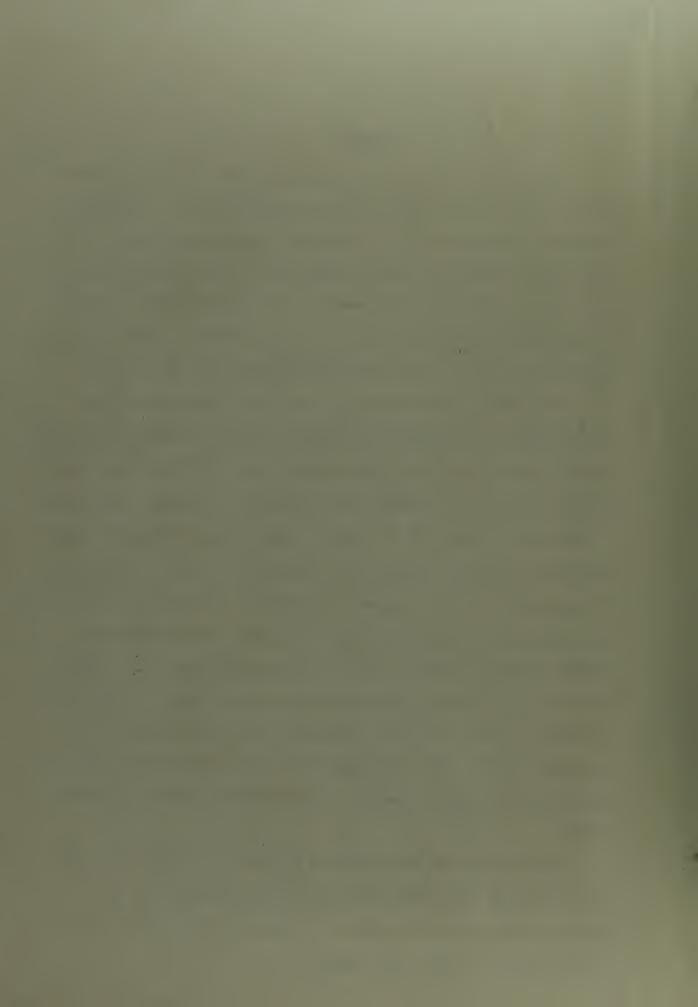


SUMMARY

The failure of turbojet aircraft propulsion units to accelerate rapidly to high thrust operation when emergencies arise in slow speed flight has restricted their use in aircraft applications, and has also concentrated considerable attention upon their acceleration characteristics in an effort to produce better results. This thesis presents a method of computing the acceleration of a particular turbojet by making use of complete performance curves of the component parts of the turbojet.

The method presented here does not permit computation of the acceleration for a particular operating condition as determined by those variables usually considered independent; namely, (1) Flight conditions of velocity, density, pressure, and temperature, (2) Engine rotor speed (3) Fuel rate of flow, and (4) Tail cone area ratio. Computation using these four independent variables was originally attempted in preparation of this thesis. However, extreme complication in the computations dictated that turbine inlet temperature and air mass rate of flow which are normally dependent variables, should be considered independent. Fuel rate of flow and tail cone area ratio are therefore considered dependent. Therefore, in order to match a particular operating condition, it is necessary to make a family of computations for various turbine inlet temperatures (constant for each set) over a range of assumed air mass flows.

Computations have been performed for the mestinghouse X19B axial flow turbojet to illustrate application of the method and to show qualitative and quantitative effects of variation of tail area ratio, at two different turbine inlet temperatures.



INTRODUCTION

approach, a rapid increase in thrust is imperative. However, current turbojet engines are notoriously slow to accelerate from low to high thrust conditions. Although a large volume of information is available concerning equilibrium running conditions of turbojets, comparatively little has been published concerning acceleration. Accordingly the purpose of this paper is to develop a method of computation of acceleration, and of thrust during acceleration of a turbojet engine; and further, to ascertain qualitative effects on acceleration and thrust of variation of tail area ratio and turbine inlet temperature.

SCOPE

The basic method developed in this analysis is general and may be applied to any turbojet operating condition. However, this method does not encompass thrust augmentation devices such as afterburning.

Application of the proposed method is contingent upon complete experimental performance data for the compressor, combustion chamber, and urbine, as separate units, and in addition upon knowledge of diffuser and nozzle efficiencies. The method is based upon the assumption that the steady state performance data can be applied instantaneously, even under non-stationary running conditions. Hence the accelerations are considered as "slow" changes, and although this assumption is probably valid, the final justification must come from comparison with tests.



The computations presented are restricted to a single flight condition. For this flight condition effects of variation of rotor speed from idle to military rating, and of tail area ratio from 0.8 to 2.0, are evaluated at two different turbine inlet temperatures.



PROCEDURE

In order to compute the acceleration of the rotor of the turbojet engine it is necessary to know both the power required to drive the compressor, W_c , and the power output of the turbine, W_t , The excess power, neglecting power required for the accessories, is then the power available for the acceleration of the rotor. In operation, the magnitude of these powers is determined by the independent variables: (1) Flight conditions (2) Rotor speed, n, (3) Fuel rate of flow and (4) Tail area ratio, $A_c/A_{5\rho}$ However, for the purpose of calculation, the independent variables have been chosen to be: (1) Flight conditions (2) Air mass rate of flow (3) Turbine inlet temperature, T_4 , and (4) Rotor speed, leaving fuel rate of flow and tail area ratio as dependent variables. This is completely explained in the discussion.

The first step used in calculating T_c and T_t was to find the entrance conditions of the compressor. Flight velocity of the engine was assumed to be 100 mph. Sea level standard conditions of density, pressure and temperature were assumed. Then assuming isentropic flow through the diffuser, a plot of T_t , and T_t , wersus air mass flow, a. was made by the use of a Mollier diagram. (See Figure 11) A particular rotor speed was then assumed. Values of mass flow from the lower limit of the compressor stall to the upper limit of the critical flow were chosen. The compressor flow parameter, T_t and speed parameter, T_t were calculated for each mass flow, and then the compressor pressure ratio T_t , the compressor power coefficient



 Ω_{c} , and the compressor temperature ratio, T_{3}/T_{2} , were found from Figures 13, 14, and 15. With these values it was possible to determine:

a)
$$P_3 = P_2 \left(\frac{P_3}{P_2} \right)$$

b) $T_3 = T_2 \left(\frac{T_3}{T_2} \right)$
c) $W_c = \Omega_c P_3 A_c \alpha_3$

P4 was assumed to be 0.98 P3. (See Figure 12). T4 was then taken as the value desired: (2000°R for one case and 2200°R for the second case). From these data it was possible to compute

$$\mathcal{T}_{t} = \frac{D_{t} n}{a_{4}}$$

$$\mathcal{T}_{t} = \frac{\dot{m}}{P_{4} A_{t}} \sqrt{\frac{RT_{4}}{2}}$$

The turbine performance charts, Figures 16, 17, 18, and 19 were entered and from them was obtained:

a)
$$\frac{P_5}{P_4}$$

c)
$$\frac{T_s}{T_4}$$

From these the following were computed:

a)
$$P_5 = P_4\left(\frac{P_5}{P_4}\right)$$

c)
$$T_5 = T_4 \left(\frac{T_5}{T_4} \right)$$

d) $\frac{P_6}{P_5} = \frac{P_6}{P_5}$

a)
$$\frac{P_6}{P_5} = \frac{P_0}{P_5}$$

(Unless Pss/Po exceeds the critical-then see

Appendix II)



With this information the nozzle chart Figure 20 was entered and M₆, T₆/T₅, and A₆/A₅ were determined. From these data $T_6 = T_5 \left(\frac{T_6}{T_5} \right)$ and

The acceleration of the turbojet rotor was then computed as

$$\dot{n} = \frac{W_t - W_c}{4\pi^2 n I_p}$$

and the thrust

for values of M_6 less than one. When P_5/P_6 exceeds the critical pressure ratio then the pressure at the nozzle outlet, P_6 , exceeds atmospheric pressure, P_6 , and a pressure term in amount $A_6(P_6-P_6)$ is added to the thrust. Under this condition, $M_6=1.0$, and the expression for the thrust becomes:

Where P_6 is determined from the known magnitude of P_5 and from that pressure ratio, P_5/P_6 , which makes the value of M_6 equal to unity.

From the plots of the data obtained from the above calculations (See Figures 1, 2, 4, and 5) it was then possible to compute the acceleration time for the rotor by taking time increments of the order of a half a second and making the computation in a step by step process.



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RESULTS

PROCEDURE FOR COMPUTATION OF TURBOJET ACCELERATION

1. Given the following atmospheric flight and diffuser conditions:

2. Assume the independent variables n and m.*

3. Compute
$$M_{z}$$
, T_{z} , and P_{z} :
$$P_{z} = P_{o} \left(\frac{1 + \frac{N_{-1}}{2} M_{o}^{2}}{1 + \frac{N_{-1}}{2} M_{z}^{2}} \right) \left(\frac{\eta_{d} \delta}{\delta^{2} - 1} \right)$$

$$T_{z} = T_{o} \left(\frac{1 + \frac{N_{-1}}{2} M_{o}^{2}}{1 + \frac{N_{-1}}{2} M_{z}^{2}} \right)$$

$$P_{z} = P_{o} \left(\frac{1 + \frac{N_{-1}}{2} M_{o}^{2}}{1 + \frac{N_{-1}}{2} M_{z}^{2}} \right) \left(\frac{\eta_{d} \delta}{\delta^{2} - 1} \right)$$

$$M_{z} = \frac{\dot{m}}{A_{z} P_{z} V \delta^{2} R T_{z}}$$

4. Compute non-dimensional compressor parameters Γ_c and Γ_c : $\Gamma_c = M_2 \sqrt{\frac{\delta'}{2}}$ $\Gamma_c = \frac{D_c \, n}{\Omega_c}$

Enter compressor performance charts for the particular turbojet under consideration (See Figures 13, 14, and 15). From these

charts determine:

- a) Compressor pressure ratio, $\frac{\rho_3}{\rho_2}$
- b) Compressor power coefficient, Ω_c
- c) Compressor temperature ratio, $\frac{T_3}{T_2}$

6. From the above data determine the following:

a)
$$P_3 = P_2\left(\frac{P_3}{P_2}\right)$$

c)
$$T_3 = T_2 \left(\frac{T_3}{T_2}\right)$$



7. Enter combustion chamber performance chart for the particular turbojet under consideration and determine pressure ratio P_4/P_3 . (See Figure 1...)

$$P_4 = P_3 \left(\frac{P_4}{P_3} \right)$$

8. Compute Γ_{t} and G_{t} :

$$\Gamma_{t} = \frac{\dot{m}}{P_{+}A_{t}} \sqrt{\frac{RT_{4}}{2}}$$

$$G_{t} = \frac{D_{t}n}{\Omega_{t}}$$

Note that I_t involves introducing another independent variable, I_4 .

For analysis of this subject see discussion.

- 9. Enter turbine performance charts for the particular turbojet under consideration (See Figure 16, 17, 18 and 19). From these charts determine:
 - a) Ps P4
 - in) It
 - c) $\frac{T_5}{T_4}$
 - a) M₅
- 10. From these data compute the following:

a)
$$P_5 = P_4\left(\frac{P_5}{P_4}\right)$$

c)
$$T_5 = T_4 \left(\frac{T_5}{T_4}\right)$$

11. Compute nozzle pressure ratio

$$\frac{P_6}{P_5} = \frac{P_0}{P_5}$$

12. Enter nozzle performance chart (See Figure 20) and determine M_6 . T_6/T_5 and A_6/A_5 . Each installation will have a different chart depending upon nozzle geometry. However, for short tail pipes and



small area ratios flow is nearly isentropic. When nozzle efficiency is known, the foregoing quantities may be determined by computation similar to those for the diffuser.

13. From these data compute the following:

$$T_6 = T_5 \left(\frac{T_6}{T_5} \right)$$

$$V_6 = M_6 \sqrt{8'RT_6}$$

14. Turbojet acceleration

$$\dot{n} = \frac{W_t - W_c}{4\pi^2 n I_p}$$

15. Turbojet thrust

a) When
$$M_6 < 1.0$$
:
$$F = \dot{m} \left(V_6 - V_0 \right)$$

b) then
$$P_{55}/P_{6}$$
 exceeds the critical pressure ratio and $M_{6}=1.0$.

 $F = \dot{m}(a_{6}-V_{6}) + A_{6}(P_{6}-P_{6})$

(See Appendix II)

Application of this method to the particular cases selected for demonstration produced the ultimate results shown in Figures 1, 2, 5a, and 3b. The effects of tail area ratio and rotor speed on thrust and on acceleration are shown in Figures 1 and 2, which were derived from Figures 7-10. Figures 3a and 3b show effects of rotor speed and tail area ratio on time required to accelerate to a particular engine speed.



DISCUSSION

The method of computation proposed in this paper is simple in form, but the calculations are lengthy and the method requires some practice to estimate quickly the proper range of mass flows to select for the calculations. This is to be expected since the mass flow is not, in fact, an independent variable. However it has been chosen as such for the purpose of ease of computation. The independent variables in the unsteady state condition are: (1) Flight conditions (velocity, density, temperature and pressure) (2) Rotor speed (3) Fuel mass rate of flow and (4) Tail area ratio. (This contrasts with the steady state or equilibrium condition where the rotor speed is a dependent variable).

In the method outlined in this paper the fuel mass rate of flow has been replaced as an independent variable by the use of a constant turbine inlet temperature. The independent variables used then are:

(1) Flight conditions (2) Air mass rate of flow (3) Turbine inlet temperature and (4) Rotor speed. Since the flight conditions have been held constant throughout the series of computations made here, there remain only three independent variables.

As a starting point in the calculation a rotor speed and turbine inlet temperature are selected. Then for each air mass flow chosen, values of tail area ratio, acceleration, and thrust are obtained. This makes tail area ratio, acceleration, and thrust a function of the air mass rate of flow. Other sets of calculations may be obtained by varying the rotor speed and the turbine inlet temperature and repeating the procedure.



The geometric configuration of the unit imposes certain natural limitations upon the selection of the air mass rate of flow. If A is chosen too low, the compressor operates in a stalled conditions. This is obviously undesirable. Under certain conditions of higher mass flow a Mach number of unity is reached at some point in the unit and a condition of critical flow exists due to sonic velocity in the turbine nozzle throat. This occurs when the value of the exceeds a critical value (In this case 0.482) and is clearly shown in Figure 10. Under certain conditions of high A when the critical flow limit is not reached, a value of A may be chosen so high that the stagnation pressure in the tail pipe is less than the atmospheric pressure. This obviously is a physically impossible condition and occurs when too high a value of A is selected. The fallacy does not become apparent in the calculations until the point of entry into the nozzle chart (Figure 20), when the tail area ratio appears to be something "greater than infinite".

The turbine inlet temperature is controlled directly by the mass rate of fuel flow into the combustion chamber. However, since the calculation of the turbine inlet emperature is a process involving the combustion efficiencies and the heating value of the fuel it was not considered to be within the scope of this report to carry out these calculations. The assumption was therefore made that a sufficient amount of fuel was consumed in order to provide the required T_4 . In order to ascertain the effects of turbine inlet temperature variation, calculations were for a T_4 of 2000° R which is approximately the maximum



allowable for continuous operation of the Westinghouse X19B (1960 $^{\circ}$ R) and for a T₄ of 200 $^{\circ}$ F higher.

Exact information regarding pressure drop across the combustion chamber was not available until after calculations were completed. The estimate used in these calculations $(\Delta P/P_3 = 0.02)$ was subsequently found to agree reasonable well with data from tests made on the combustion chamber of this unit by Westinghouse. (See Figure 12).

No allowance has been made in this analysis for mass of fuel added, air leakage between compressor and turbine, or power required for accessories. The effects of these small quantities tend to compensate each other.

Figures 1, and 2 show the effects on acceleration and thrust of variation of rotor speed and tail area ratio at a particular turbine inlet temperature. Comparison of these two charts shows that temporarily exceeding the peak continuous allowable temperatures of the mestinghouse X19B produces a slight increase in acceleration at low rotor speed, but also introduces the danger of operating within the compressor stall.

It is interesting to note that acceleration may be obtained only at the expense of thrust. This further aggravates the problem of thrust requirements under emergency conditions.

Convergence of lines of constant tail area ratio as their magnitude increases shows that increasing tail area ratio above 2.0 produces only slight gain in acceleration.

Figures 3a and 3b show the effects of rotor speed and tail area ratio on time required to accelerate to a particular engine speed.

Because the accelerations is a function of the small difference between



large turbine and compressor powers, accuracy of the results is subject to question, and should be verified by experimental data.

Pertinent extracts from a report on flight tests of a Mies unit conducted by the Clenn L. Martin company are slown in Figure 6. Since manual control of fuel flow was used in these tests, a constant turbine inlet temperature could not be mintained. An effort was made, however, to follow a procedure which would produce maximum acceleration. No rational estimate of turbine inlet temperatures used in the tests is possible. However, since maximum acceleration was their goal, it is presumed that these temperatures were at or near the maximum allowable (1960°R) during most of the run. Tail area ratio was held constant, but at a value not specified. However, estimates derived from a photograph of this installation indicate the tail area ratio was approximately 2.0. Acceleration times, then, would be roughly comparable to those shown in Figure 3a, for a tail area ratio of approximately 2.0. The agreement is close enough that quantitative values may be considered roughly correct, and the qualitative effects may be considered reliable.



CONCLUSIONS

- 1, Acceleration characteristics of a turbojet may be computed by the method presented in this report, provided adequate experimental data for component parts is available.
- 2. Results obtained through use of this method agree closely enough with experimental data so that quantitative values obtained may be considered roughly correct, and qualitative effects may be considered reliable.
- 3. This method is not applicable for prediction of a schedule of acceleration for a turbojet under actual operating conditions until extensive calculations have been made over a complete range of turbine inlet temperature and flight conditions.
- 4. Increasing tail area ratio increases acceleration but with diminishing effect as tail area ratio gets larger.
- 5. Increasing peak temperature increases acceleration at all rotor speeds.

 The relative increase is much greater at higher speeds.
- 6. Increasing peak temperature tends to induce earlier compressor stall.
- 7. Acceleration of a turbojet under any condition is slow when compared to a reciprocating engine.



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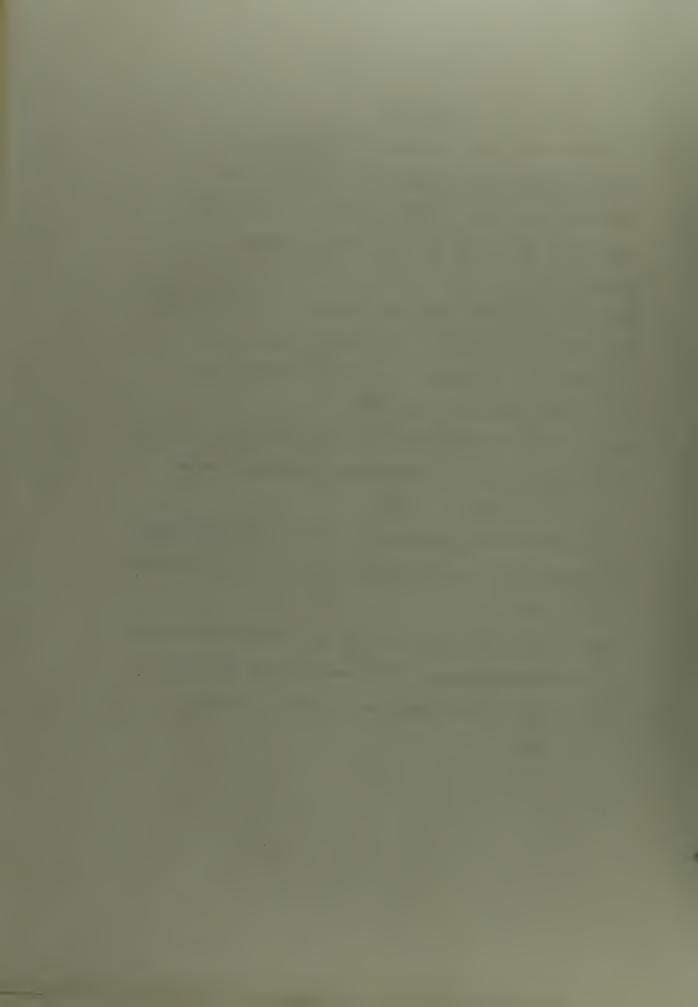
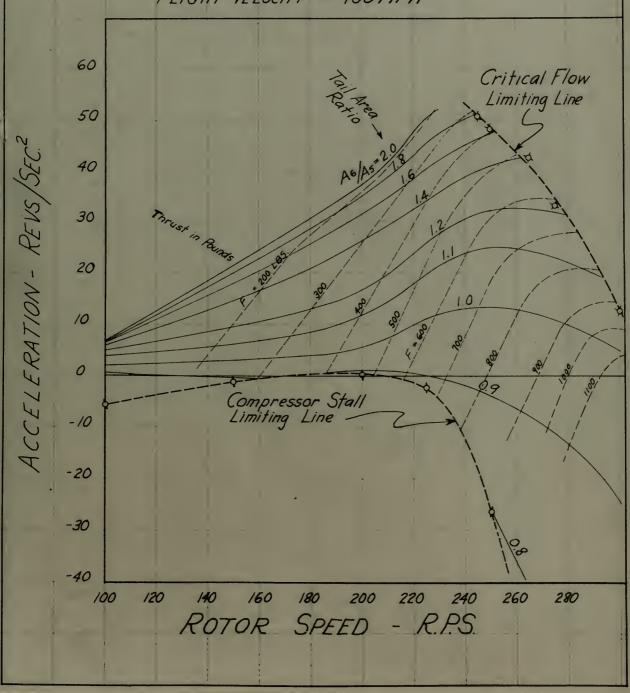


FIGURE 1 TURBOJET ROTOR ACCELERATION VS. TAIL AREA RATIO AT VARIOUS ROTOR SPEEDS

TURBINE INLET TEMPERATURE - 2000°R SEA LEVEL STANDARD CONDITIONS FLIGHT VELOCITY - 100 MPH



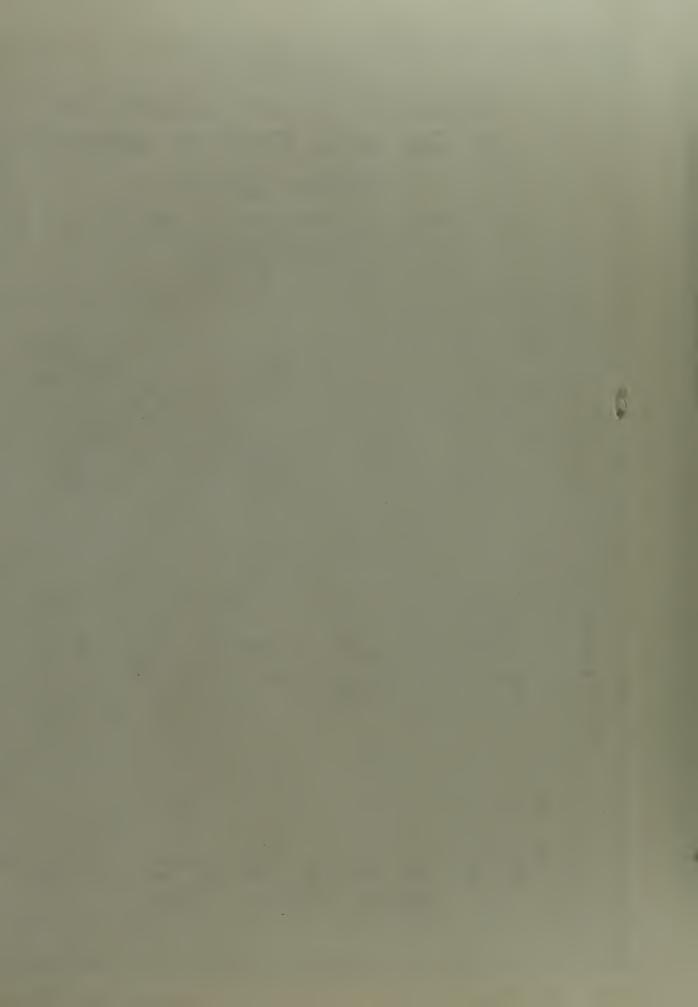
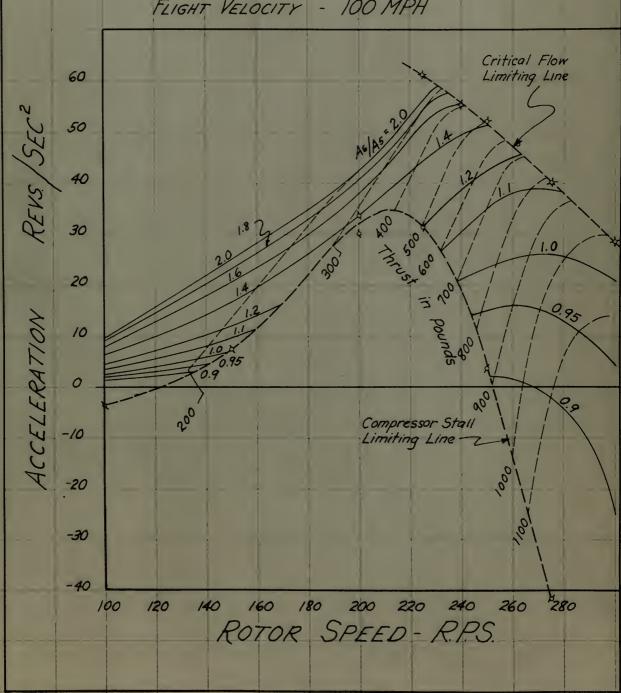
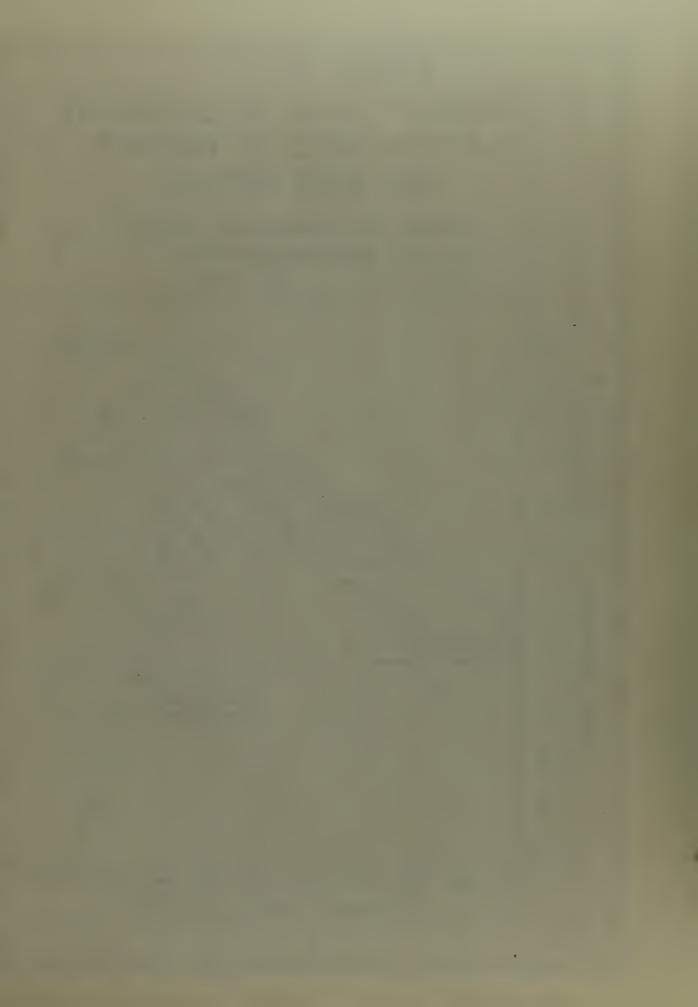
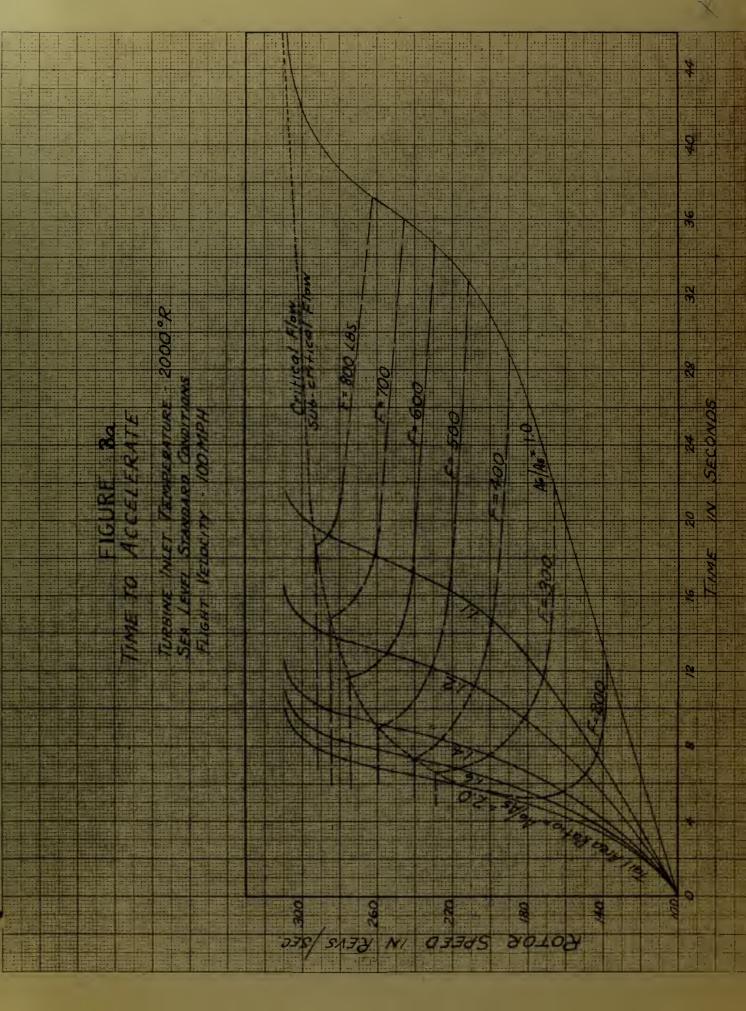


FIGURE 2 TURBOJET ROTOR ACCELERATION VS ROTOR SPEED AT VARIOUS TAIL AREA RATIOS

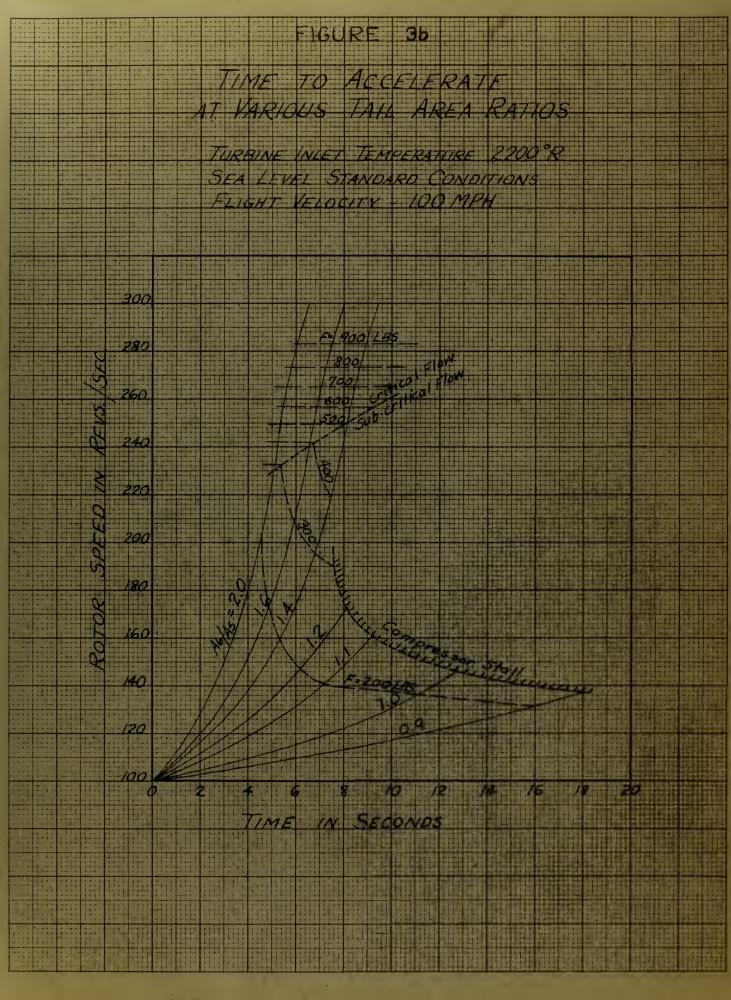
TURBINE INLET TEMPERATURE 2200°R. SEA LEVEL STANDARD CONDITIONS FLIGHT VELOCITY - 100 MPH



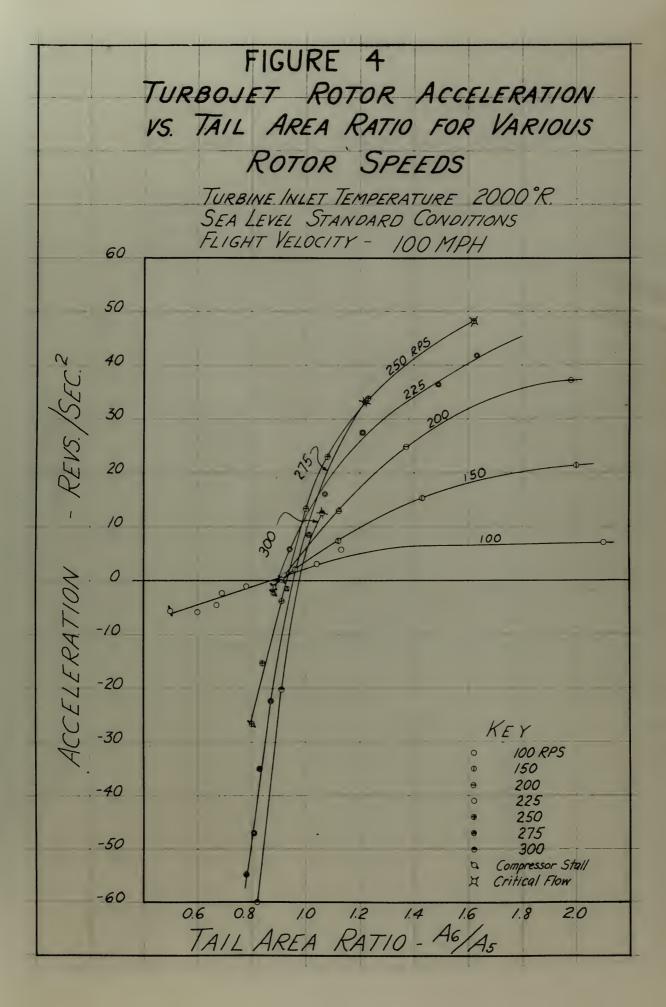












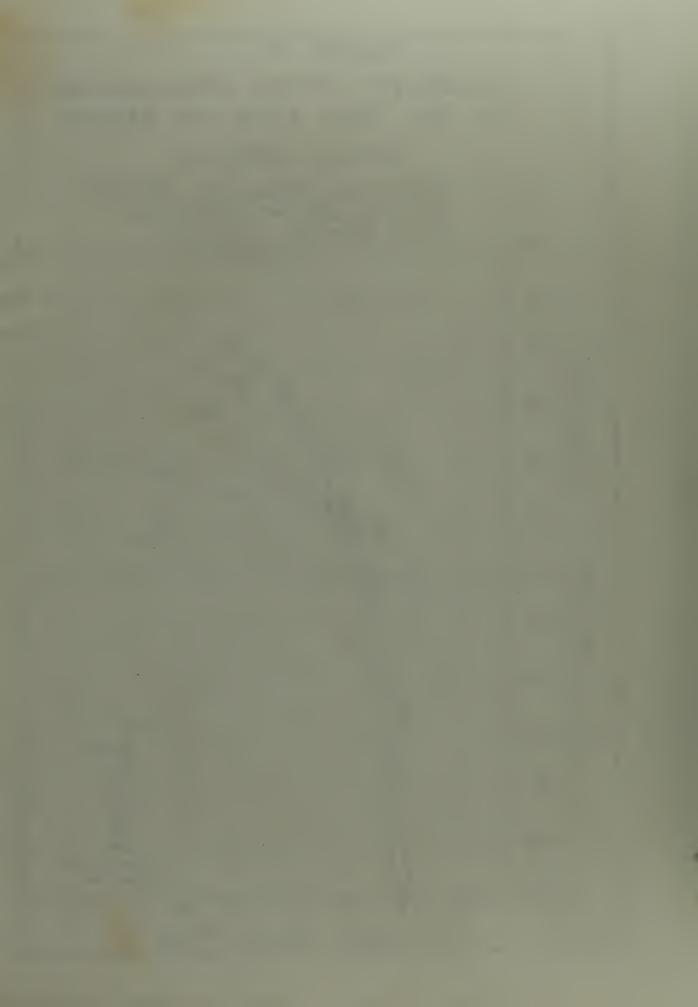
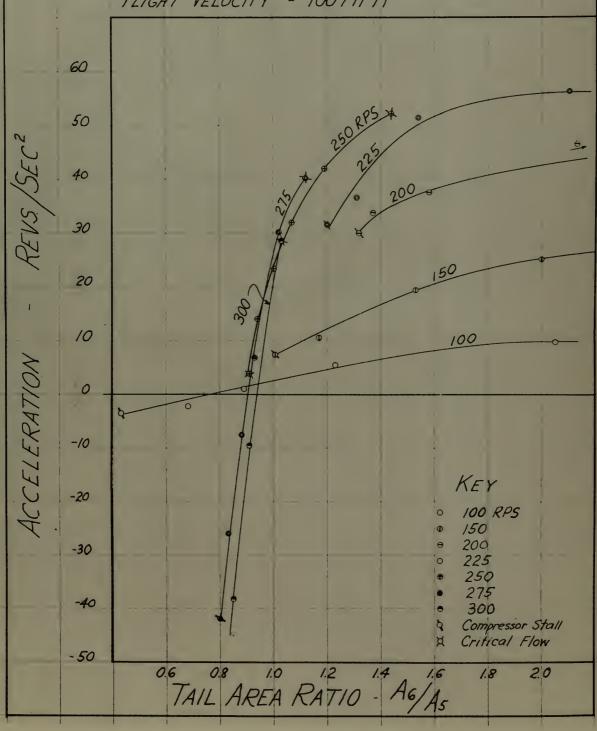
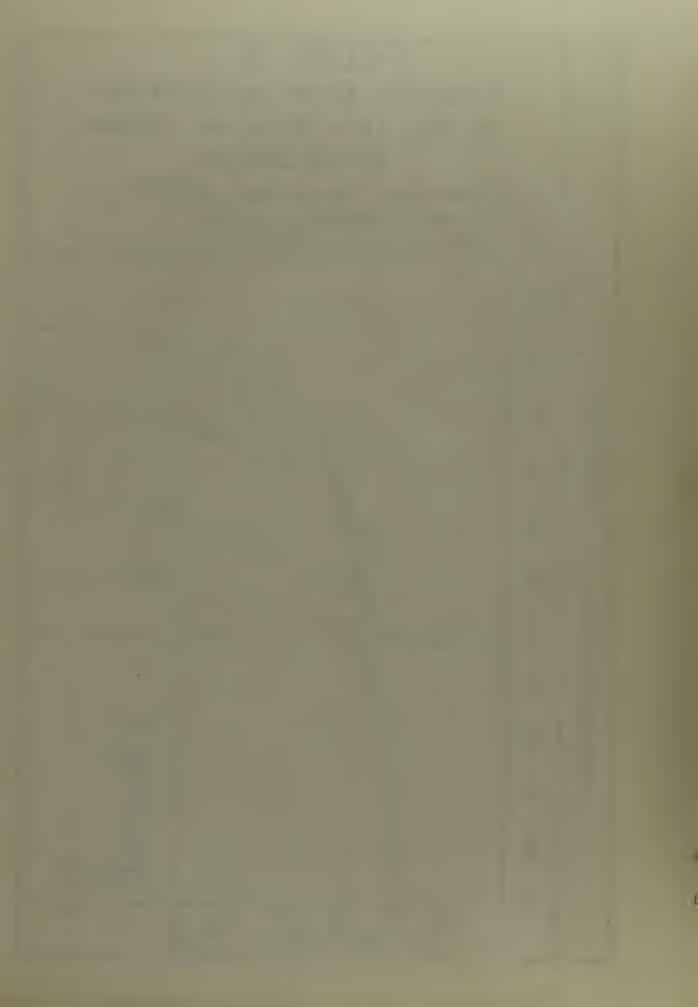
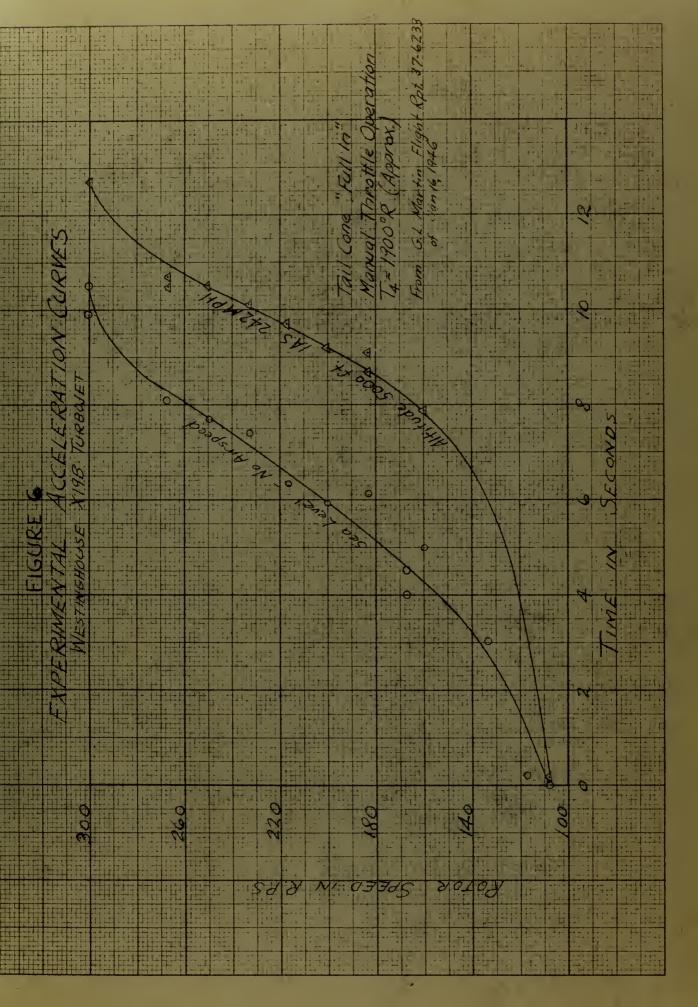


FIGURE 5 TURBOJET ROTOR ACCELERATION VS. TAIL AREA RATIO FOR VARIOUS ROTOR SPEEDS

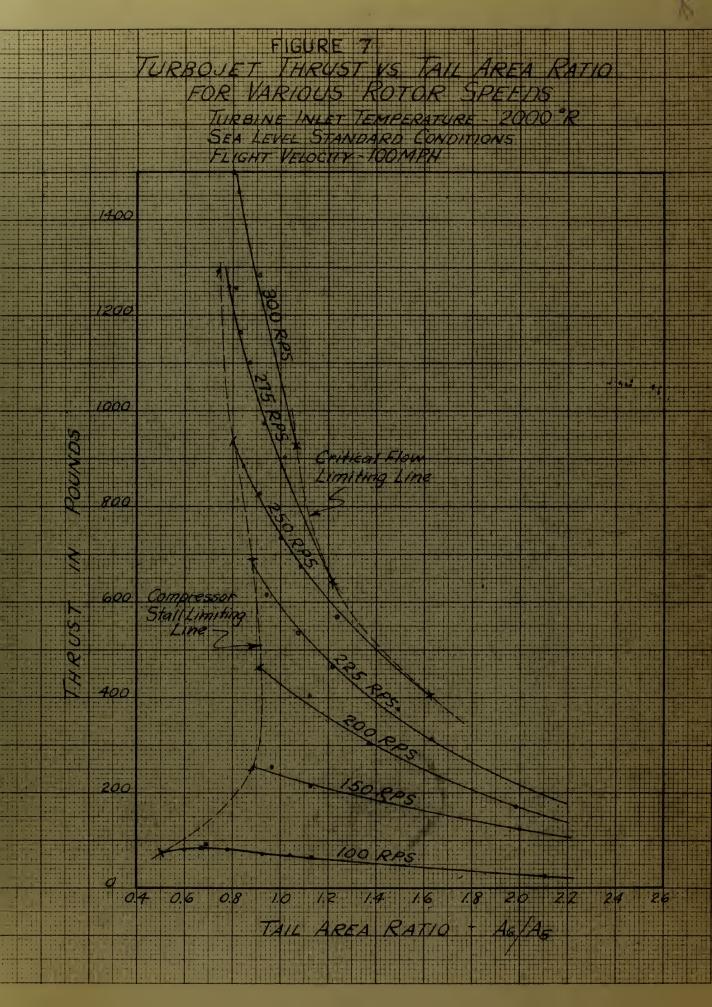
TURBINE INLET TEMPERATURE 2200°R SEA LEVEL STANDARD CONDITIONS FLIGHT VELOCITY - 100 MPH



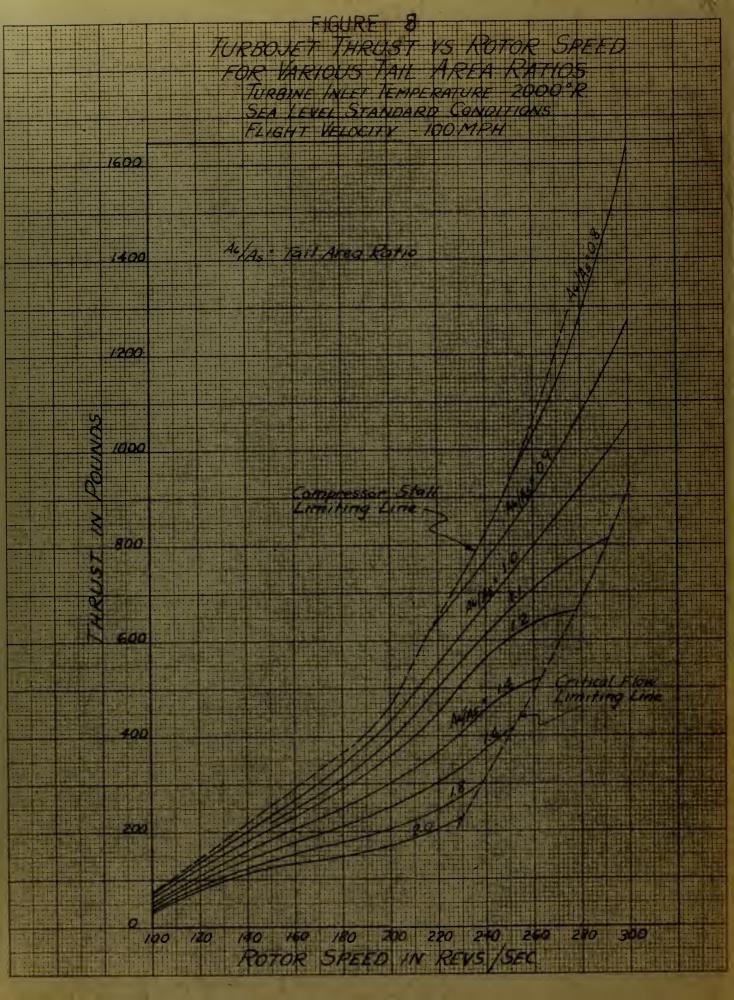




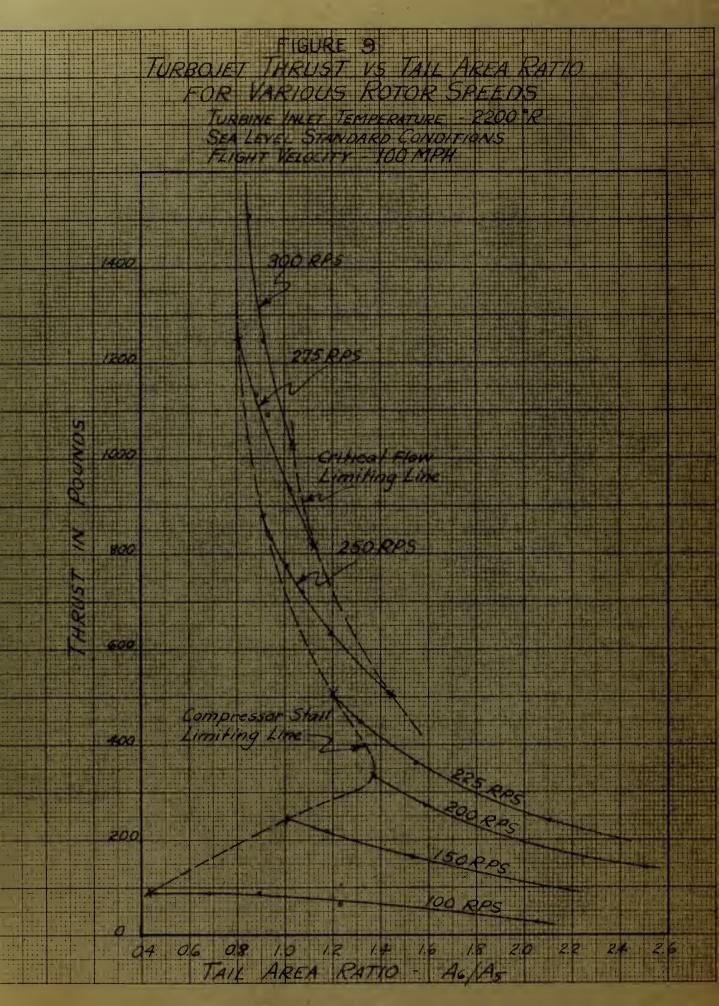




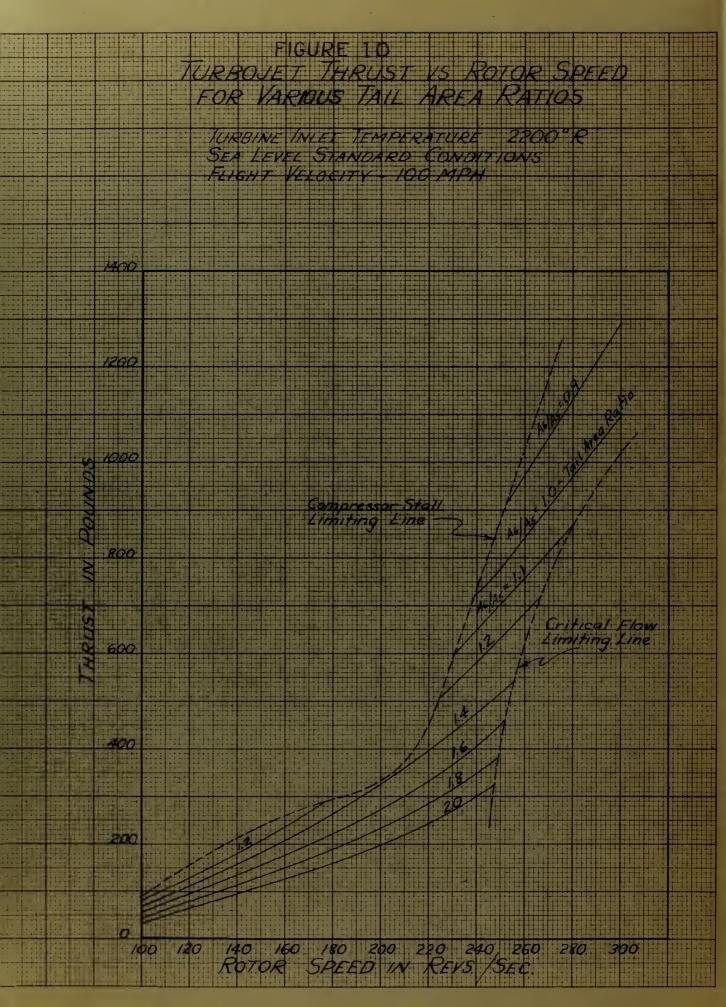




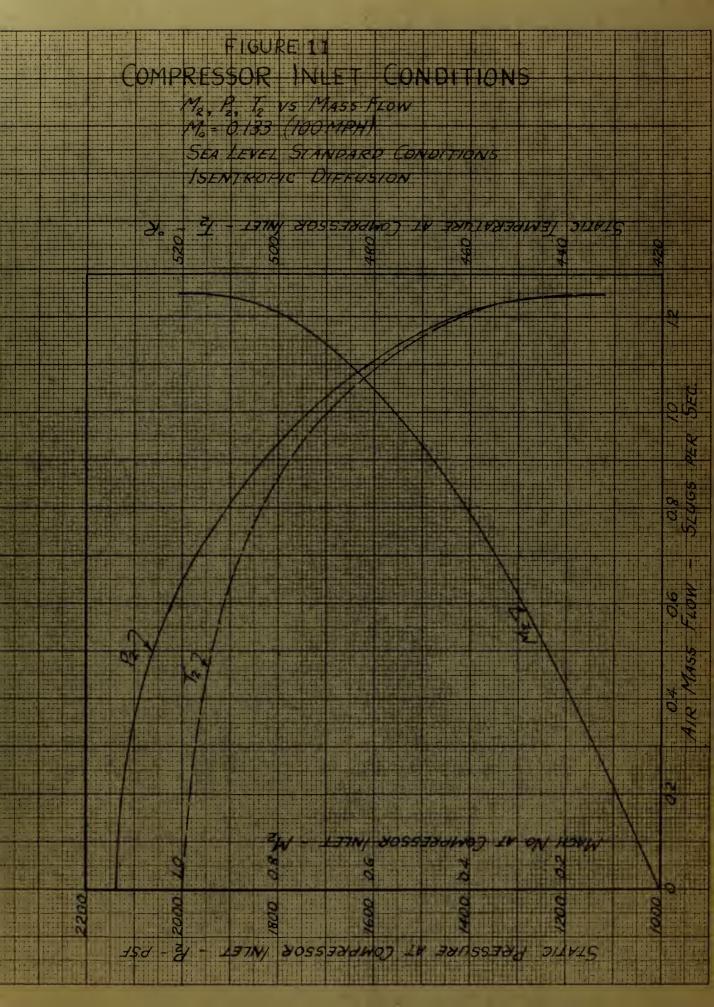








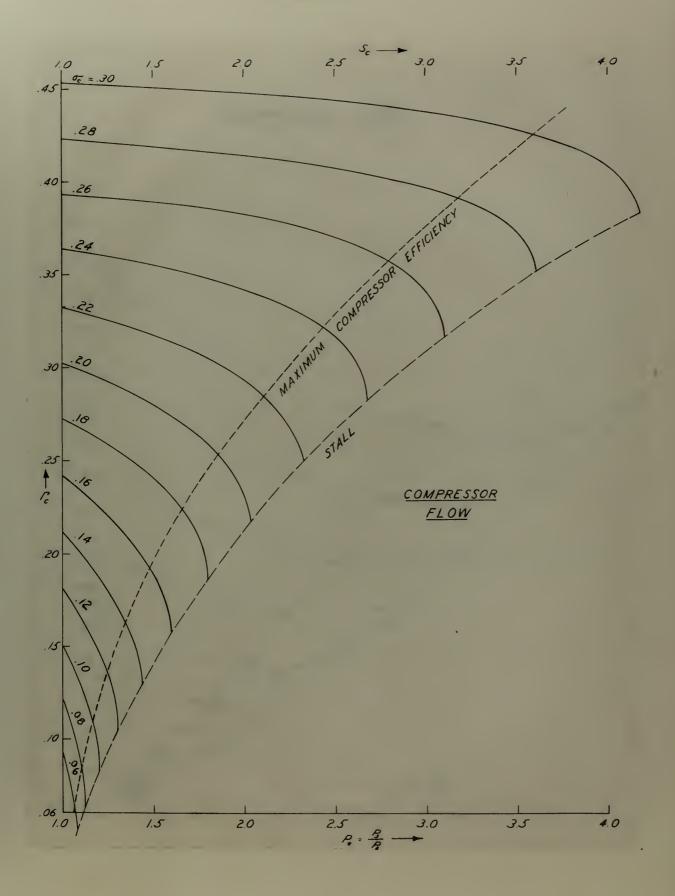






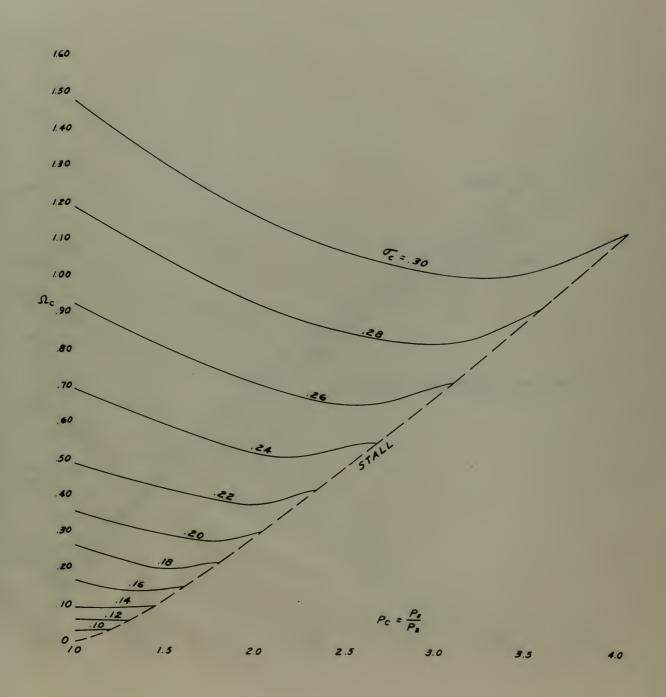
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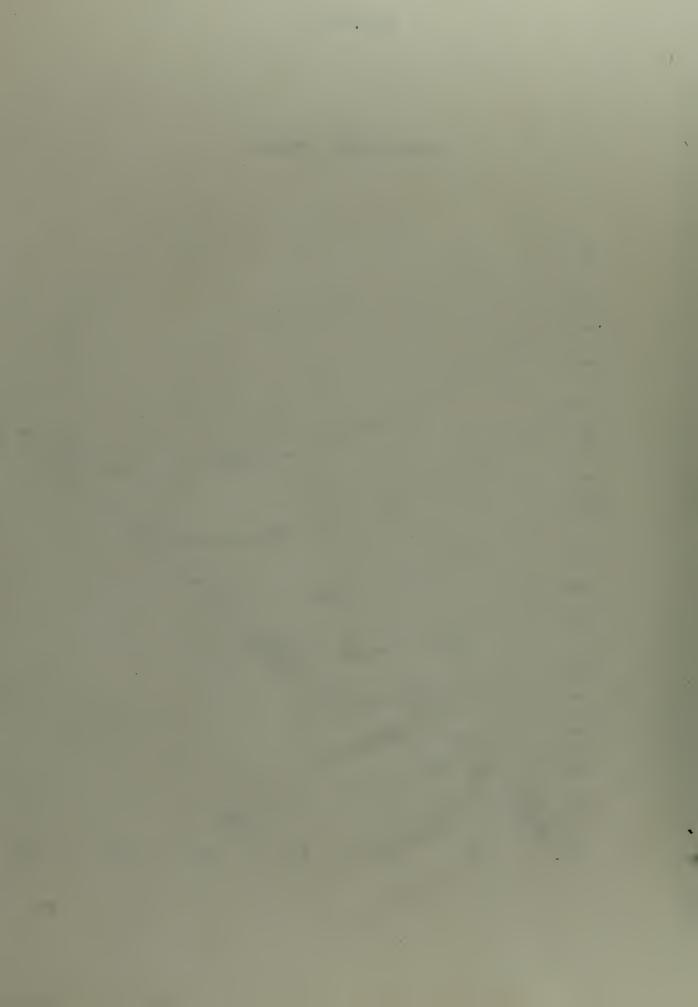




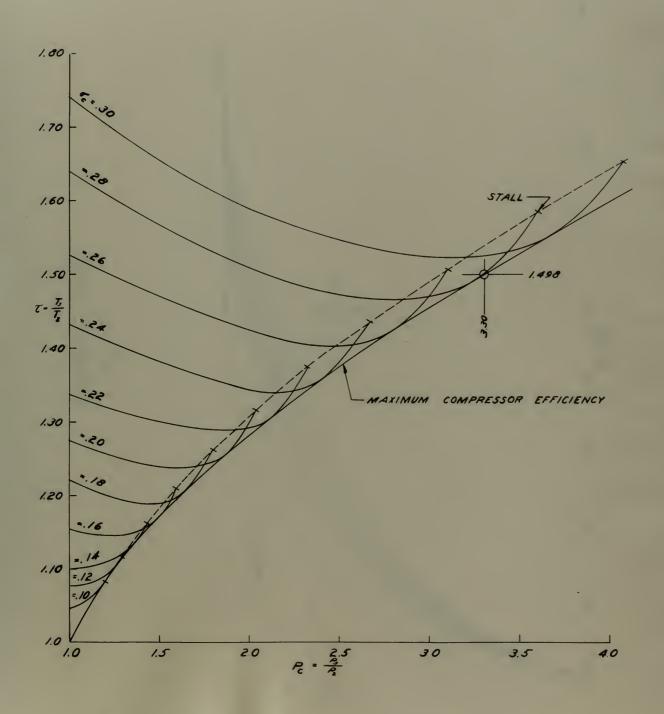


COMPRESSOR POWER

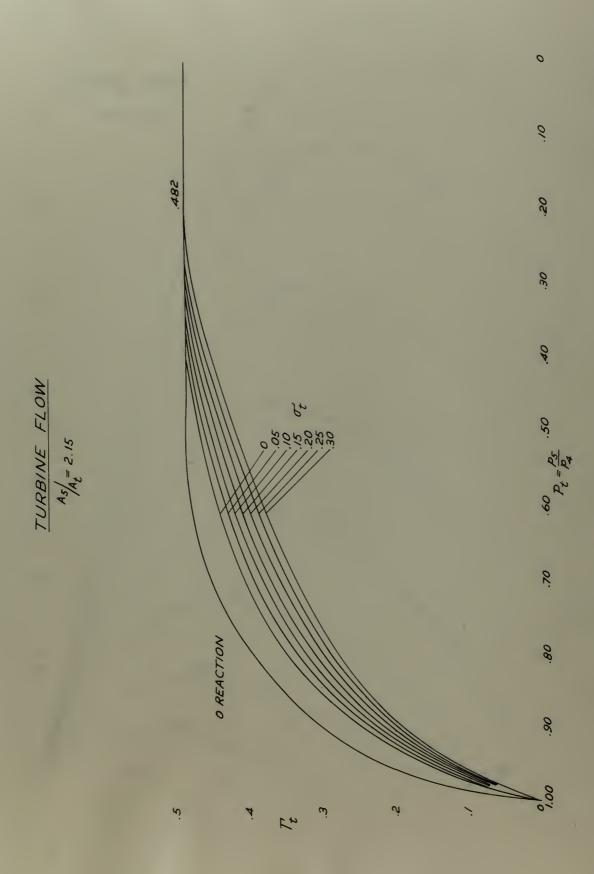


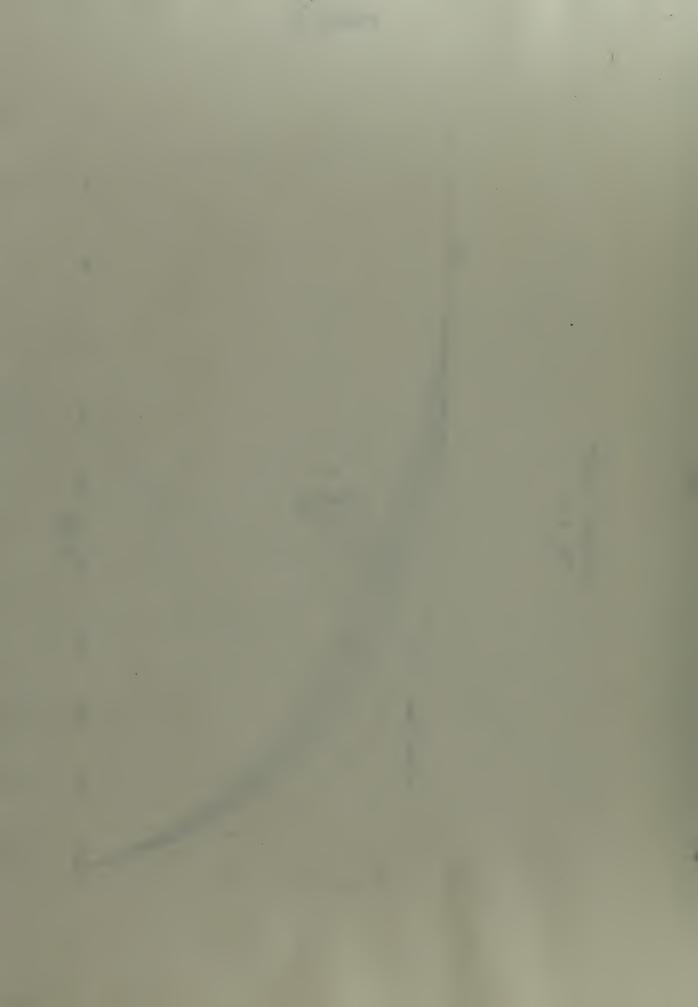


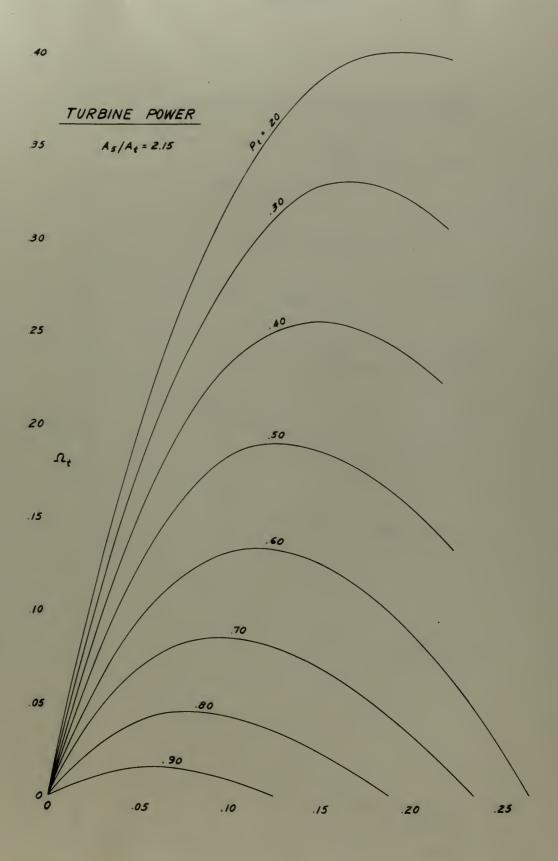
COMPRESSOR TEMPERATURE RISE



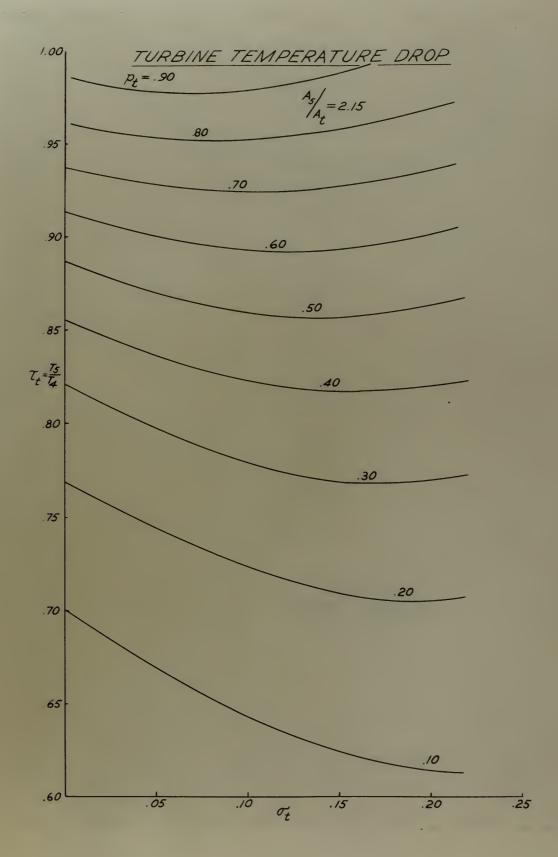




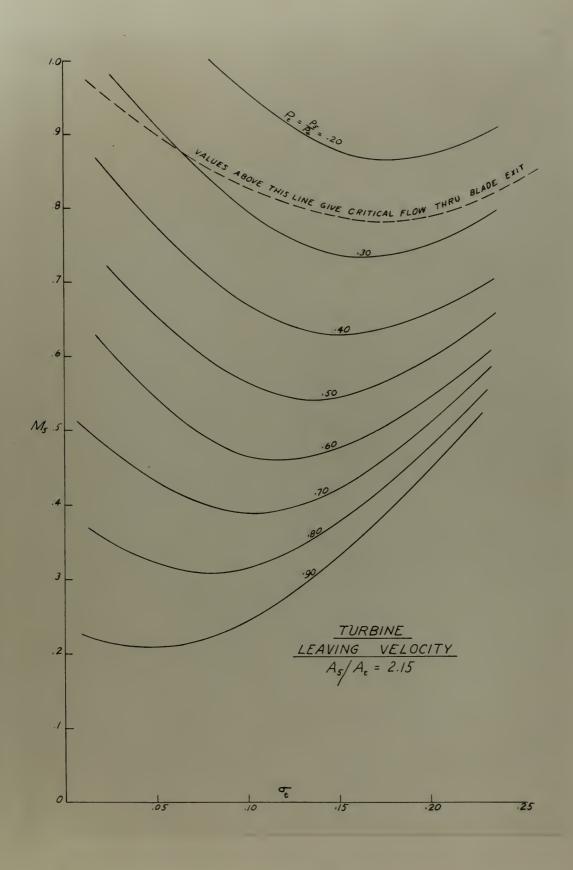






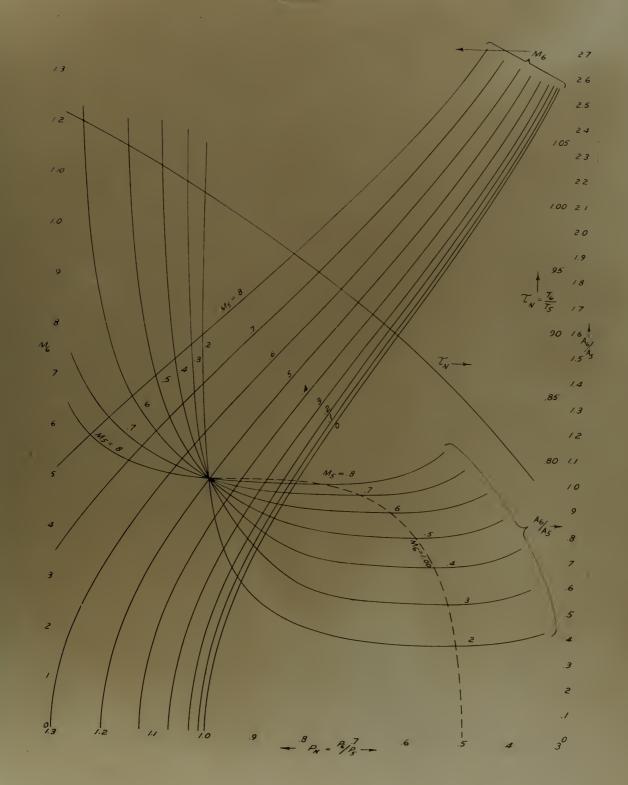














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APPENDIX III

DETERMINATION OF MEAN GAMMA

Compressor: Mean F = 620°R

Fivid : Alm

Turbine: inlet T 1500°R +5 2400°R

Outlet T 1200°R +0 2000°R

Average T 1350°R +0 22:0°K

Fluid: Fuel/Air Katio C. 01 to 0.03

Temp.	Fuelnin	CE	K_	5-
1050°K	0.01	6.570	1/16	1.5
1350°R	0.0=	6750	1710	1.57
2-20°K	0.01	1590	1110	1.50
12 30° K	0.03	1110	1,10	1.31
			Avg. =	1.33

Note: VILLES of of and K although from Minkel





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